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14. ABSTRACT
This report documents the process, analysis of alternatives and decisions made by an undergraduate team from the University of Missouri-Columbia in designing a radio controlled (RC) aircraft to perform a specific sensor related mission in the 43rd AIAA Joint Propulsion Conference Student Design Challenge. The challenge was to design an integrated propulsion and power system capable of sustaining flight for video surveillance of ground targets while generating additional power to drive a power consuming device. Aircraft design constraints included a single propeller vehicle with a maximum take-off weight of 15 lbs; a commercially available airframe with wing span 80-82 in. and fuselage length 62-67 in; a power consuming device with a 28 volt input and integrated into the airframe or mounted externally on the airframe. The amount of power consumption and degree of continuous video surveillance were measures of merit for the design challenge.

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Unmanned Aircraft System; Lithium-Polymer batteries; student design challenge; electric motor; small UAV; video surveillance

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Objective:

Develop and demonstrate innovative power and propulsion concepts

Approach:

Selected undergraduate student teams will participate in the 43rd AIAA Joint Propulsion Conference Student Design Challenge to build and fly a radio controlled (RC) aircraft using specified airframe parameters to perform a specific sensor related mission. The design challenge is an integrated propulsion and power system capable of video surveillance of ground targets while generating power to drive a power consuming device. Data and video will be down linked to a ground station that will calculate the amount of power consumption and display continuous surveillance.

Progress:

Year: 2007 **Month:** 10 **Final**

This report documents the process, analysis of alternatives and decisions made by an undergraduate team from the University of Missouri-Columbia in designing a radio controlled (RC) aircraft to perform a specific sensor related mission in the 43rd AIAA Joint Propulsion Conference Student Design Challenge. The challenge was to design an integrated propulsion and power system capable of sustaining flight for video surveillance of ground targets while generating additional power to drive a power consuming device. Aircraft design constraints included a single propeller vehicle with a maximum take-off weight of 15 lbs; a commercially available airframe with wing span 80-82 in. and fuselage length 62-67 in; a power consuming device with a 28 volt input and integrated into the airframe or mounted externally on the airframe. The amount of power consumption and degree of continuous video surveillance were measures of merit for the design challenge.

43th AIAA Joint Propulsion Conference Student Design Challenge

Final Design Report

from

University of Missouri – Columbia "The Flying Tigers"

1.0. Executive Summary

The goal of this project is to design, build, and fly an unmanned aerial system (UAS) at the 43rd AIAA Joint Propulsion Conference Student Design Challenge. The objective for the competition is to design an integrated propulsion and power system that will maximize the amount of electrical power generated or provided on-board an unmanned aircraft (UA) to be flown on a specific mission profile that includes video surveillance. Power not used for the propulsion system or on-board electronics will be supplied to a power consuming device (PCD). Constraints on this design include specifications of the airframe and the required on-board electronic devices to monitor telemetry and provide surveillance. After comparing the weight, excess power generation capabilities, and several other factors (such as stealth) while taking into consideration the prescribed mission profile, an electrical based system was chosen using Lithium-Polymer (Li-Po) batteries and a brushless out-runner electric motor to power the propeller. Excess power available for the PCD is 1200 W, corresponding to a current draw of 42.8 amps while maintaining the required 28 V.

1.1. Design Outline

This project will involve designing, building, and flying an unmanned aerial system (UAS) at the 43rd AIAA Joint Propulsion Conference Student Design Challenge.[1] This competition is sponsored by the American Institute of Aeronautics and Astronautics (AIAA) and the U.S. Air Force Research Labs (AFRL). It will be held on June 14-16, 2007 at Wright-Patterson Air Force Base near Dayton, Ohio. The objective for the competition is to design an integrated propulsion and power system that will maximize the amount of electrical power generated or provided on-board an unmanned aircraft (UA) to be flown on a specific mission profile that includes video surveillance.

The mission profile includes an un-assisted take off and climb to 250 ft altitude with a takeoff roll of under 300 ft., ideally under 200 ft. The aircraft will then have one minute to enter a course after climb out. The course will consist of two 180 degree turns at least 700 ft apart. Aircraft will enter the course between 200 and 250 ft of altitude and maintain level flight within this range. The aircraft will then complete as many laps as possible while conducting surveillance and a power consuming task. The

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[1] http://www.afrlstudentdesign.com/index.html

maximum flight duration is 10 minutes. Only laps where ground targets on the course are identified using the video surveillance system will be counted.

The designed systems will supply power to all onboard electronics, propulsion system, and an electronic power consuming device (PCD). Power supplied to the PCD will be monitored in real-time and serve as the basis of the scoring for excess power generation. Constraints on this design include specifications of the airframe and the required on-board electronic devices to monitor telemetry[2] and provide surveillance. Because of these constraints the design of the airframe will be limited to modifying an almost-ready-to-fly radio controlled airplane[3] to accommodate the PCD and other onboard electronics. The entire Unmanned Aerial System (UAS) will also include a pilot, a manual remote control unit, UA on-board flight control systems, and a laptop that will monitor the telemetry and surveillance in real-time.

2.0. Management Summary

The Flying Tigers student design team consists of student studying Mechanical Engineering (ME) and range from freshman to graduate students. Several of the team members are pursuing an Aerospace Emphasis in their ME Engineering degrees. The layout of the team was split into three main sub-teams: 1) airframe/ control, 2) electronics, 3) and powerplant/ propulsion with the inclusion of thermal management into the propulsion team. The design chairman managed the three team leaders and communicated with them directly. The team leader then organized and managed their particular tasks. It should be noted that each group member work throughout across the spectrum of the project. They were not limited to their designated title that was simply their emphasis. The idea was to keep each group member involved in the entire project so that everyone was on similar levels of understanding. The role of the design chairman was to ensure quality work and timely scheduling. The goal was to adhere to the proposed schedule contained in the initial design proposal submitted in November 2006. Figure 1 shows an organizational chart which illustrates the break-down of each team member's general duties under competition guidelines:

[2] http://www.eagletreesystems.com/Plane/plane.html

[3] http://www.sigmfg.com

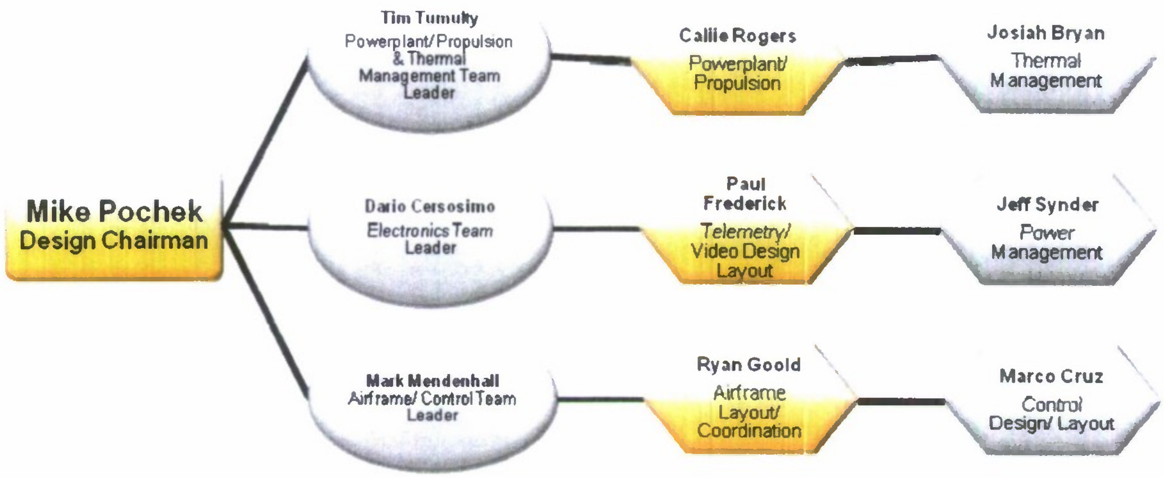


Figure 1: The Flying Tigers Organization Diagram.

Each sub-team worked on their respective aspect of the design, but as previously stated each group communicated with each other in order to completely integrate the airframe to fulfill the required mission. For example the main task of the airframe/ control team is to construct, fabricate, and/or modify the airframe to support the system internally. Additional analysis and wind tunnel tests were done on the airfoil and a quarter-scale model, respectfully. The electronics team main objective was to achieve an understanding of electronic systems that were going to be placed on-board. It turned out that a voltage regulator needed to be constructed in order to maintain certain systems' integrity.

2.1. Calendar and Schedule

Throughout the design process, a tracking calendar was used to guide the progress of the design; this was the main task of the design team chairman. Figure 2 describes the schedule used throughout the design process. Included are both the predicted and actual progression timeline, detailed in blue and red respectfully. Notice that the grant was not awarded until the first of February 2007, thus pushing back the desired timeline back about one and one-half months. In doing so, the testing and flight scenario tasks had to be reduced to still make the competition deadline. It can also be shown that even though the initial start of the project was delayed the progression of the design maintained the same trend, just shifted forward chronologically and moved closer together. Brief descriptions of each progressive stage of the design are presented in the following subsections.

2.1.1. Proposal – *Design Approach and Conceptual Design*

This part of our project has been completed. We have looked at several options during this phase and have finalized our design concept based on the calculations made during this task.

2.1.2. Final Report – *Assembly Phase and Testing Phase*

The final report task is a continuous process. As calculations and decisions are made they are recorded and will be included with the final report. The final report task includes two major components, the assembly and testing phase. The final report assembly task includes the calculations made in the design approach and conceptual design task as well as the revisions required during the assembly of the aircraft and related systems. The testing portion of the report task includes the time it will require to document our progress in testing and updating the design as we test, modify, and complete the project.

2.1.3. Assembly

Component Purchasing

This is the initial phase of the competition. Purchasing the component is of course an ongoing task and probably will be until the fly-off. Besides the needed components, most of the purchasing is on an as-needed basis. The major components include the airframe, electric motor, Li-Po motor battery packs, PCD battery packs, etc. Some of the unnoticed purchases are things such as tools, flight simulators, training expenses, organizers compartments, etc. Components that see a high-stress life have been accounted for and are monitored carefully. In some cases a duplicate component will be purchased as a back-up. A full list of components purchased is to be included in later sections.

Airframe Modifications

Coordination between all three sub-teams has to come together on the modification to the airframe. Each sub-team has their needs and requires space in the airframe. The airframe/ control team leader coordinates that effort, to ensure that the airframe is configured to accommodate all systems and sub-systems required to maintain safe and efficient flight.

Power Consumption Device (PCD)

The PCD consist of heating element wire that acts as the resistance for the dissipation of power. A brushed motor controller is used to control the flow of power into the wire. This can be done from the ground and will be monitored through the telemetry system. Testing of this system requires great care due to the temperatures that the wire might obtain if not handled properly. A more detailed description of the design and fabrication of the PCD system are presented in a later section of this report.

Propeller & Motor configuration

The motor, controller, and basic size and pitch of appropriate propellers have been chosen. A test bench was build to measure the force of each propeller combination. Test data for these tests are in later sections.

Electrical Component Wiring

Electrical wiring is important to the sustainability of the design. A system that is poorly wired has more potential to short circuit the system and destroy the components involved. A clean schematic is also in further sections; this diagram lays out the mapping of the electrical system and gives a good idea of where components are and where they reside in the airplane.

2.1.4. Testing – Airframe Simulation, Unweighted & Weighted

During the initial testing phase work will be done on the ground to further improve the design. The goal of this static testing program will be to maximize the power output of the on-board battery system without overstressing the airframe. Some components such as the electronics, while they will still need to be tested, do not affect the flight performance except by the weight that they add. Testing and simulation of the components on the ground will provide a better understanding of their behavior in the air.

Other components such as the motor, propeller, power generation device, and the PCD must be tested in the air to determine optimum flight parameters. Observing how the components interact with each other is the only true way to integrate the system to achieve optimal performance. The aircraft will be tested in a minimum and maximum weight conditions during the flight tests. The goal of the flight testing is to determine the flight envelope of the aircraft in the competition environment and provide data for optimized performance.

2.1.5. Scenario Building – *System Integration & Flight Manual*

During this task the competition course will be duplicated and flight testing will take place while simulating the variables that are anticipated. A flight manual be created and will consist of an array of different scenarios. These scenarios will be built around how the competition flight tests are scored and will include various combinations of weight, power, aircraft configuration, and performance altitude calculations, including altitude density calculations.

2.1.6. Final Testing

The flight manual will be an integral part of the UAS at the competition. Final testing will use the manual to aid the pilot and the crew, allowing them to quickly and easily change the aircraft configuration to get the maximum performance from the aircraft for the conditions at the time of flight. During the competition the results of the other teams will also be analyzed. The team will then be able to use optimization software on-site to determine the best aircraft configuration to achieve the optimum balance of take-off performance, speed, and power output to achieve the highest possible score.

2.1.7. Flight Video

We are required to deliver a video of our aircraft in flight to AFRL on or before June 8. This is required before we are allowed to participate in the competition.

2.1.8. Fly-Off

This task is the competition itself and our ultimate goal. The fly off will take place on June 14-16 in Dayton Ohio at Wright-Patterson Air Force Base.

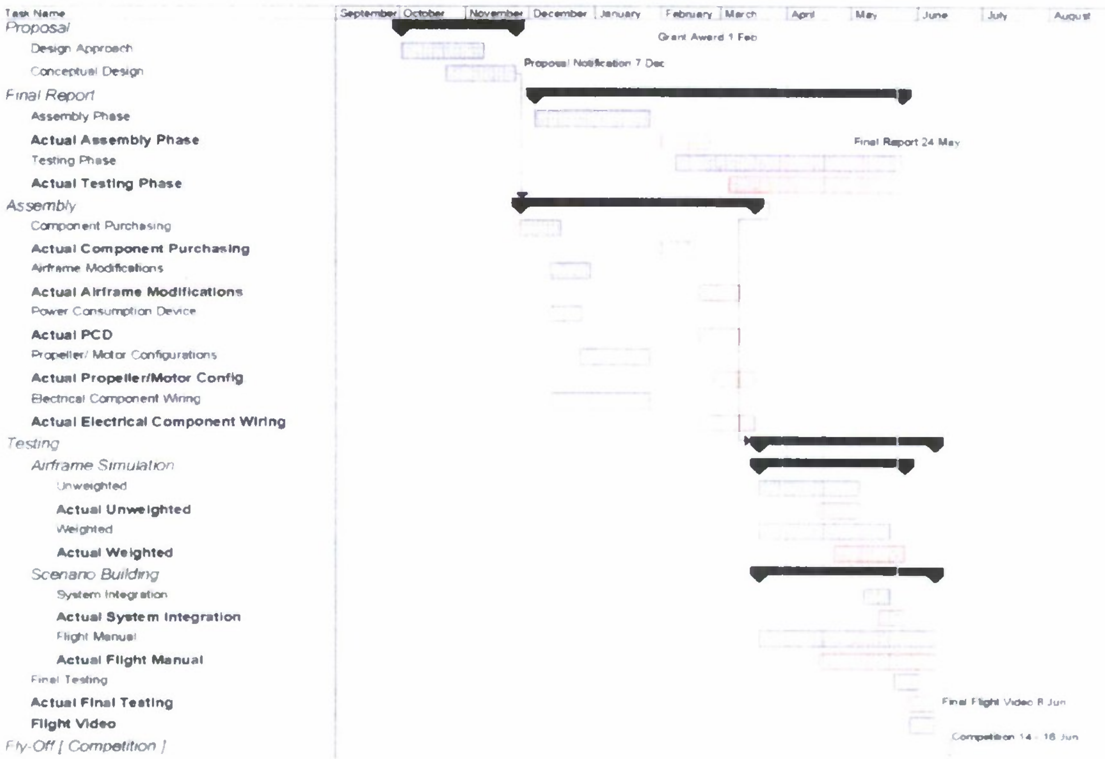


Figure 2: Gantt chart for Project Scheduling and Management. Note that the blue blocks represent projected timelines and the red blocks represent actual events. Several milestone dates are mention around the approximate date along the timeline.

In our submitted proposal we stressed testing as an important entity in the success of the UAS. Even though the testing blocks were reduced from the proposed timeline, we still feel that our testing plan accounts for a majority of the flight problems. Given that this is our first competition as a newly started organization, there were some start-up delays that were not considered, such as trouble finding lab space, lack of an existing work experience, and the start-up cost for all the tooling and design materials. Invaluable knowledge has been gained and will thus lay the groundwork for future design team future advance our student organization.

3.0. Conceptual Design

In order to conceptualize a project of this nature a thorough understanding of the rules is imperative. Additionally, a deep knowledge of the submitted proposal is needed, given that the submitted proposal was the basis for acceptance into the competition. The concept of the design is based off the competition design statement. One must consider all options at this point; weighing the pros and cons of each design option and from that make a decision based on the one that fits the overall design objective the best. An accepted method of comparing design paths is a system using figures of merit (FOM's). Another method uses a decision matrix; these two methods are virtually the same. However in our application it was decided that the decision matrix offers a better understanding of what would provide the optimal conceptual design. Table 2 displays the contrast between different systems and used to make design decisions based on marginal performance. This method uses a value system for different categories that rank each design path to see which concept is the best over all the categories. The categories are easily derived from the problem statement and are a good starting point for the design. This is very similar to Chapter 3 of our submitted proposal.

3.1. Design Problem Statement

This project will involve designing, building, and flying an UAS at the 43rd AIAA Joint Propulsion Conference Student Design Challenge.[4] This competition is sponsored by the American Institute of Aeronautics and Astronautics (AIAA) and the U.S. Air Force Research Labs (AFRL). It will be held on June 14-16, 2007 at Wright-Patterson Air Force Base near Dayton, Ohio. The objective for the competition is to design an integrated propulsion and power system that will maximize the amount of electrical power generated or provided on-board an UA to be flown on a specific mission profile that includes video surveillance.

The competition is scored on four main parameters. These parameters include the written report score, takeoff distance, number of laps, and power delivered to PCD. Each is equally worth 25% of the total competition score. Table 1 shows a break-down of each of the four parameters along with the formula for scoring.

Table 1: Final Competition Scoring Break-Down.

Written Report Points	Takeoff Distance	Number of Laps	Power Delivered to PCD
$WRP \cdot 0.25$	$R_k \cdot \frac{MTO}{CTTO} \cdot 0.25$	$\frac{CTNOL}{MNL} \cdot 0.25$	$\frac{CTTPO}{MTPO} \cdot 0.25$
WRP= Written Report Points	MTO= Minimum Takeoff Distance CTTO= Current Team Takeoff Distance	CTNOL= Current Team # of Laps MNL= Maximum # of Laps	CTTPO= Current Team Total Power Output MTPO= Maximum Total Power Output

[4] http://www.afrlstudentdesign.com/index.html

Each of the scores is based on the ratio of the best score in the category compared to your current score in the category.

3.2. Alternative Designs

In order to accomplish the design objectives for this UAS of maximum excess power generation, several alternatives to provide UA propulsion and power were examined. Three primary propulsion systems were examined which included both fuel powered and electrical powered motors. Various alternatives to power generation were also examined. The details of these analyses and a decision matrix for these concept evaluations are outlined in this report.

Three main options for propulsion and electrical power generation were considered: 1) a turbine, or 2) an internal combustion (IC) engine, with either coupled with an electric generator, and 3) an electric motor powered by a battery pack. In any configuration excess power could be produced by additional batteries, a fuel cell, photovoltaic cells, thermoelectric modules or a combination of these systems. However it was found that additional batteries gave far greater power for the amount of weight than all the other electrical generation alternatives.

The first option is a turbine. As ducted fans are not allowed in this competition the only way to use a turbine would be to drive a propeller through a gear reduction system. It would also be difficult to duct the output of a turbine in such a way as to produce no thrust because of the type of small turbines that are available and the limited space in the airframe. The complexity of this configuration is not practical even though the power generation capabilities of a turbine connected to an electric generator would be more than an IC engine.

The second option for main power is an IC engine. Two different types of IC engines are commonly available, 2-stroke and 4-stroke. 2-stroke engines have a distinct advantage as they are more powerful for the same weight and size. Also, an IC engine connected to an electrical generator has the capability of producing power for long periods of time. However, the amount of electricity that is able to be generated is dictated by the size of the engine. Generators that would produce a significant amount of electricity for short periods of time require a relatively large engine. Two disadvantages to 2-stroke engines are the difficulty in starting and operating a glow plug engine, and the oil blow-by. This oil tends to coat the airframe, can interfere with the on-board electronics, and in this design, possibly foul the video camera lens.

The third option is purely electric-based power. Very recent advances in battery and electric motor technology have made the most advanced electric systems comparable in power and efficiency compared to IC engines for short duration flights and lightweight aircraft. Brushless electric motors have efficiencies of over 70% compared with around 60% for the best conventional electric motors. In addition, out-runner brushless electric motors are now commonly available. In these motors the majority of exterior case rotates along with the output shaft. These motors are even more efficient than the traditional in-runner motor where only the shaft rotates and have efficiencies of over 85%. Also, out-runner motors turn more slowly for a given power output. This eliminates the need for a gear reduction gear system that is required for in-runner motors when used in UA applications; this further increases the efficiency of an out-runner based design and significantly improves the reliability of the system. To power this motor Li-Po batteries are currently the most advanced commercially available batteries as they are lighter and smaller than similarly powered Nickel-Cadmium (NiCad), Nickel-Metal-Hydride (NiMH), or Lithium-Ion (Li-ion) batteries.

A significant consideration in this UA design is the specified 10-minute duration of the mission. The shorter the flight time the greater advantage an electrical system has as most of the weight is in the batteries instead of the motor as is the case with an IC engine. As flight times are increased the advantage goes to IC engines as added fuel is much lighter than additional batteries. Initial calculations show that the power output per weight for a 10 minute flight and size UA required for this competition is nearly equivalent for a 2-stroke IC engine or an out-runner brushless electric motor and Li-Po battery combination. This does not include the addition of any power generation device which would only significantly increase the advantage of a battery based system over an IC engine.

The calculations made in order to choose a propulsion system, main power source, and potential supplementary power sources are outlined in this section. First, propulsion systems were evaluated using an engine size recommended for the chosen airframe. It is planned to use a larger engine in the final design but these calculations were used as a baseline comparison. All the options were reduced to two primary choices, a 2-stroke IC engine and a brushless out-runner electric motor. The weight of all the components of these two systems, including fuel tank, fuel and throttle control for a 0.46 cu in glow plug IC engine and batteries for a 12 minute flight and electronic controller for a 670 rpm/volt electric out-runner brushless motor, was 30 oz. and 30.1 oz respectively. The component weights for each system are detailed in Table 2. This gave the opportunity to select the best propulsion system based excess electrical power generation capabilities.

Table 2: Engine Weight Comparison

IC Engine		Electric Engine	
Part Description	**Weight (oz)**	**Part Description**	**Weight (oz)**
0.46 cu. in. 2 stroke + muffler	16	670 RPM/V out-runner	10
fuel tank	6.5	LiPo batteries	19.2

fuel	6		electronic controller	0.9
throttle control servo	1.5		**Total**	30.1
Total	30			

Since the thrust-to-weight ratio is nearly equal for the two propulsion systems examined, the electrical power generation capabilities of these systems were then calculated. A company that specializes in UAS generators, Sullivan Products , usually mates a generator capable of producing 500 W of continuous power with a 9.69 in^3 engine. However, the engine sizes recommended for our airframe are in the 0.40 to 0.60 in^3 range. Electric generators made by Sullivan for these size engines weigh only 5.5 oz but are only capable of producing only 3.84 W. In comparison a 4.7 oz Li-Po battery pack is capable of producing 140 W for 10 minutes. Also, this additional battery pack would not reduce the power output to the propeller as would an electric generator. This clearly gives the advantage to an all electric battery powered propulsion system which we have chosen for our design. Based on initial calculations, the power available from batteries will largely be limited by the weight lifting capabilities of the airframe. It has been estimated that the excess power available from batteries should be approximately 1680 W.

These calculations were then used to fill out a decision matrix as shown in Table 3. A decision matrix is commonly used method to rank alternative design concepts.[5] This matrix shows various decision criteria and their relative importance in percent. The alternatives were then listed and given a rank from 1-10 for each criteria, based on the relative performance of each alternative. The satisfaction index in percent was then calculated. These values are the sum of all the criteria importance percentages times the rank for that alternative divided by ten.

Table 3: Conceptual Design Decision Matrix Mated With FOM

			Alternatives		
	Criteria	**Importance**	**Gas Turbine**	**IC Engine**	**Electric Motor**
1	Flying Performance	12%	8	6	7
2	Weight	35%	5	5	5
3	Power Generation	35%	3	2	9
4	Stealth	2%	1	3	8
5	Turn Around Time	3%	8	8	6

[5] http://en.wikipedia.org/wiki/Decision-matrix_method

6	Ease of Use	5%	5	5	9
7	Reliability	5%	4	6	9
8	Cost	3%	4	7	3
	Total Importance	100%			
		Satisfaction	45.90%	42.30%	70.70%

Importance = % out of 100%

Alternatives are ranked 1-10, 10 being the best

Satisfaction = Σ_{1-8} (Importance *Alternative Ranking / 10)

4.0. Preliminary Design

Using a blend of the tables, the preliminary design can start to take shape. This phase focuses on critical aspects of the chosen design path only and examines the analysis method to optimize the aircraft's performance.

4.1. Design Parameters and Sizing Trades

Investigation of design parameters and sizing components ensure that the aircraft's performance is fully optimized to the mission constraints and requirements. The design parameters are based off the importance of the design matrix criteria. The relative importance is a normalized percentage of the sum of criteria for the mission. Conducting trade studies on these parameters will derive the relationship between the factions of the design; this can is similar to a parametric study. Understanding which parameter changes the performance of the design the most gives an accurate portrait of the design sensitivity. This sensitivity can be partial reflected in the decision matrix which roughly maps the important performance criteria.

4.1.1. Aerodynamic Design Parameters

The competition is based on the "guts" of the airframe and not on the aerodynamic efficiency. This eliminates any aerodynamic design-work out of the competition. However this does not inhibit run tests and analysis of the existing design. This data can be used to predict the performance of the airframe during different stages of flight. Since all the design parameters are essentially set, the only controllable variable is the flow conditions the plane will see. In order to quantify this several different method were used to ensure the analysis had a reasonable precision. Below is a bullet list of the aerodynamic post-analysis conducted:

- ∴ **Numerical Analysis –** A computational fluid dynamic (CFD) software package, Fluent, was used to determine the lift coefficient and estimate the drag coefficient for flow similar to the conditions for the competition.
- ∴ **Analytical Analysis -** Equations derived from aerodynamic theory were used to again estimate the airfoil lift coefficient. This method was used to compare and contrast to the result of the numeric analysis.
- ∴ **Experimental Analysis –** The final analysis method used a scaled model of the competition aircraft. The wind tunnel tests were conducted at Saint Louis University in their low-speed wind tunnel. This method incorporated the entire aircraft with hopes of obtaining the quantified lift and drag data to aid in flight performance predictions. To get this result, similar flows need to obtain using dimensional analysis.

4.1.2. Propulsion Design Parameters

The design of the propulsion system is the heart of the competition. It has already been stated that the optimal design is that with a combination of an outrunner DC electric motor connected to Li-Po batteries. There are four main design parameters for the propulsion system. Unlike the aerodynamics of the airplane, the propulsion system requires preliminary design analysis and is up to each team as the final design choice. The four design parameters are as follows:

- ∴ **Engine Selection –** Brushed and brushless engines were considered. Brushless motors provide a greater efficiency. In order for an electric motor to out-preform a standard internal combustion engine the power density must be high. Additionally, the required power for the engine also factors into the selection.
- ∴ **Battery Selection –** The capacity, discharge rate, and weight of the batteries are the driving parameters for the battery decision. Lithium-polymer batteries offer a greater power density than equivalent nickel-cadmium (Ni-Cd) or nickel-metal-hydride (NiMH) systems in terms of power per unit weight, or power density. A capacity will need to be determined based on the duration of the flight and the power requirements from the selected engine.
- ∴ **Voltage (Speed) Controller Selection –** The selected controller needs to have the capability of supplying the peak current that the engine needs to sustain maximum thrust. This is of course based on the engine selected and requires a brushless controller. Ideally the controller also needs to be relatively efficiency to conserve on waste power. Programmable cards allow the settings of the controller to be changed given the application, this is something desired for a controller that needs to be versatile.
- ∴ **Propeller Selection –** The takeoff and cruise performance dictate the selection of the propeller. The pitch and diameter of the propeller offer different advantages and disadvantages. Due to the increased gross takeoff weight a larger diameter propeller offers great initial thrusts. The pitch

helps for slower spinning propellers, since electric engines spin at a greater RPM than standard IC engines there is no need for a high pitch propeller. A good figure of merit of the proper propeller is the efficiency of the blades when it is matched with the selected engine.

4.1.3. Power Generation (PCD) Design Parameters

Power generation based on the conceptual design consists of lithium-polymer batteries as the energy source. Both the sink and source for the design are free variables. As stated in the conceptual design section of this report, it was justified that Li-Po battery offer far greater power densities than comparable power systems, this will further be validated with actual calculate in the detailed design of the report. However, there are three design parameters that encompass the power generation/ dissipation, these parameters are as follows:

∴ **Source Selection –** Maximum energy needs to be available to the PCD in order the consume the most amount of power. To do this it makes sense to just carry all the power with you from the start. Stated from the conceptual design there are no real effective ways of generating power. The short time constraints make it unviable. Conversely, the power that is available needs to be able to dissipate its energy at a fast rate. It now becomes clear that another battery circuit provides all of those things.

∴ **Source Control Selection –** The propulsion parameters call for a brushless speed controller, which actually pulses the power to make the brushless motor turn. Both the propulsion and power generation design parameters call for battery systems; it makes sense to use a voltage controller to maintain the required system voltage of 28 V. For this a brushed voltage controller offers the perfect solution because it can be monitored and controlled from the ground to increase accuracy.

∴ **Sink Selection and Thermal Management –** Thermal management of the dissipated power becomes the driving factor: Since the power level is high for the device that is transporting all the systems. According to that the sink selection needs to be able control the heat generated from the power. In essence, the sink acts as a big resistor and dissipates the heat into the ambient surroundings. The layout of the sink needs to be such that the heat generated is in a control volume outside the airframe, given the airplane's construction material and its combustion volatility.

4.2. Mission Model

Using the design parameters as a guide, the mission can be model for the same parameters with a better understanding of the role of each component in the system as a whole. Analysis of each parameter results in best performing aircraft. The mission model can be summarized once all the parameters have been processed to then reduce the sensitivity of each parameter.

Incorporated in the model needs to be the scoring metric for the competition. The competition is basically scored off the normalized best score in that category. Therefore the best approach to achieve success in the competition is to try and do well in all factions of the competition. The formulas used to predict performance are essentially the score for the competition. Ideally, setting the normalized score will give you the best score. The detailed design expands on these issues.

4.2.1. Sensitivity Model

It can be described that the weight of each component affects the total design the most. There is a maximum weight limit of 15 lbf, and a lot of design decisions where predicated off of that boundary. For example, battery trays were considered to be the means of securing batteries, but their individual weight push the gross takeoff weight over the limit. This caused major changes in the design. The batteries were studied in detail based on calculations that will be introduced in the detail design section. They were sized off the duration of the flight and the circuit voltage limits of 28 V. The propeller selection was based off of thrust-sled tests of each propeller, the one that performed the best was chosen as possible final configuration.

4.3. Sizing

The aircraft was sizing based on the scoring parameters detailed in Table 3. The selections are further explored is further sections.

4.3.1. Aerodynamic Characterization

The aerodynamics of the aircraft have been previously designed and sized for us, and is therefore not the focus of the competition. The follow is the approach to predicted the pre-design performance of the aircraft under the weights and loads introduced to fulfill the competition mission. Numeric analysis was done on the airfoil, which is a Clark-Y, using a computational fluid dynamic (CFD) software package, Fluent. The flow characteristics were chosen to resemble those of the flying conditions at the competition as closely as possible. In addition to the numerical analysis, analytical analysis was done on the airfoil as well to compare with the numeric data. The basis for the analytic analysis is from the fundamental equations governing circulation flow around an airfoil. The thin airfoil theory was applied around the mean chamber line of the Clark-Y airfoil. The final prong of the aerodynamic analysis was wind tunnel tests of a quarter-scale model. The model was constructed of balsa and bass woods with a monokote covering. The objective of the tunnel test was to obtain similar flows to those seen by the actual airframe. Unfortunately exact flow similarities could not be obtained due to the scaling effects. Since the model is quarter-scale, a constant Reynolds Number was attempted; in order to achieve this, the flow over the scaled model needed to be four times that of the full-scale model. The required flow would have exceeded the material limits of the model as well as risked causing the attachment point to fail. As a result it was determined that the risk of trying to obtain data equivalent was not worth the risk of not getting any data at all. Figure 3 and 4 are illustrations of the constructed quarter-scale model as well as its configuration inside the wind tunnel.

Figure 3: Picture of constructed scaled model used for wind tunnel testing.

Figure 4: Picture illustrating the configuration of the scaled model in the wind tunnel test section.

If further exploration of the aerodynamic post-analysis is desired, Appendix A contains more detailed information concerning the calculations and test data. The resulting data was used to determine the takeoff performance of the aircraft along with the maximum speed and the rate of climb. The raw data displayed the L/D ratio, and maximum C_L. The results of the data are included in the detailed design portion of the report.

4.3.2. Aircraft Control

Again the control surfaces the stability derivatives have previously been determined. The only innovation that was implemented was the programmed flapper. This was programmed with the transmitter and will give us the able to takeoff with more control than without flaps, a clean configuration.

Digital servos where chosen over analog servos; this was done to improve the precision of the control throw. As well as to increase the position holding against the extra feedback on the control surfaces due to the increased weight relative to the normal aircraft.

4.3.3. Propulsion

The engine was sized off provided literature for us by AXI. This information includes sizing ranges for particular airframe configurations. Additionally the sizing includes efficiency optimization. We have chosen the size of the engine based on the power requirements for the engine to fulfill the mission and that was cross-referenced with the ideal engine size to give us an optimal engine efficiency. The engine battery systems were then designed based off that knowledge. Propeller combinations were tested on our fabricated thrust stand. The thrust stand's concept was to attach the engine to a mount that is to be secured to two different springs, each with the same displacement constant, and measure the current, power, and of course displacement of the engine at different throttle settings. Figure 5 below is a picture of the thrust stand used to conduct these tests.

Figure 5: Picture illustrates the test stand layout.

Once the motor is turned on and controlled via the transmitter, the thrust pushes the assembly along the track and the resulting displacement was measured and a final force was backed out of that value. Graphs of this data for all of the propellers tested are located in Appendix B if further exploration is desired. The chosen propeller combinations are further described in the detail design section.

4.3.4. Power Generation

The PCD battery circuit was designed around the duration of the flight and possible off-the-shelf battery systems that could be purchased. The design factored in the capacity of the batteries, their charge density, and the weight of the components themselves. Once that was determine the PCD wire was specified. The wire is typical heating element wire that is used to absorb a great deal of energy. The design incorporates the resistance of the wire and designs the length of it to match the voltage requirement for the system. Given that the voltage was set, the only thing that could be changed was the resistance of the wire; changing the resistance changes the current delivered to the wire. This is how the power generation/ dissipation systems were sized for the mission.

4.4. Predicted Performance

This aircraft uses a Clark Y airfoil and lift to drag ratio tables available for this airfoil were used to estimate maximum load lifting ability.[6] This work was done along with a numerical and analytic analysis of the Clark Y airfoil which was required for the competition final report. This work can be found in Appendix 4 and Appendix 5. Tables at appropriate low Reynolds numbers will be used to approximate the anticipated competition flight speeds. These tables are readily available as the Clark Y airfoil, which dates back to 1924,[7] has been extensively tested.

In addition, some aircraft performance calculations were made to determine the optimum aircraft configuration. The two options explored were 1) a light weight aircraft capable of faster flight but lower PCD output and 2) a heaver, slower aircraft with maximum PCD output. These two options were explored since the competition evenly weights PCD output and the number of laps achieved in 10 minutes over the prescribed course. This course is shown below in Fig. 4. Note the representative surveillance targets on this figure (smile face and the number 3). Both surveillance targets must be identified for a lap to count.

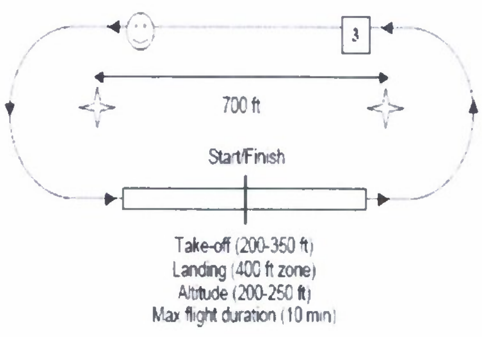


Figure 6: Conceptual Layout of Flying Tack

[6] Marchman, J.F., and Werme, T.D., Virginia Polytechnic Institute and State Univ., Blacksburg, VA, AIAA 22nd Aerospace Sciences Meeting, January 9-12, 1984

[7] http://www.hq.nasa.gov/pao/History/SP-445/ch2-3.htm

Figure 7: Aerial overlay of the flying field.

In order to determine the optimum aircraft performance configuration time to climb estimates were made and lap time calculations were done. It was estimated due to the low maximum altitude that the time to climb would minimally effect the number of laps that would be completed. This estimation was based on known performance characteristics of the same airframe which was able to climb at approximately 1000 ft per min.[8] Our aircraft will be heavier but also has a more powerful engine. Actual climb flight performance is difficult to calculate since it would require the drag characteristics of the entire airframe. These are not available without wind tunnel test data for this airframe which was not practical to obtain. Actual flight performance characteristics will be compared to the estimated and calculated flight performance parameters once flight testing begins.

Lap time calculations were made as turn rate (φ) and turn radius (R) are only dependant on bank angle (φ, in degrees) and airspeed (V). The equation for turn rate in radians per second is shown in Eq. (1) where $g = 32.2\ \text{ft/s}^2$.

$$\dot{\psi} = \frac{g \tan \varphi}{V} \tag{1}$$

For this competition the course includes two 180 degree turns or 360 degrees per lap. The time (t) in seconds to complete one full turn is given by Eq. (2).

$$t = \frac{2\pi V}{g \tan \varphi} \tag{2}$$

[8] http://www.masportaviator.com/ah.asp?ID=12&Index=0

Therefore for it actually takes more time for the turn at faster speeds. It was then calculated that at faster speeds the time gained over the 700 ft straight sections of the course will approximately be lost in the amount of time it takes to turn. For example one lap at a constant airspeed of 60 ft/s (~41 mph, a slow flying speed) will take a minimum of 35 seconds to complete at a bank angle in the turns of 45°. In comparison, one lap at a constant 90 ft/s (~61 mph, a fast flying speed) will take a minimum of 33 seconds to complete at a bank angle in the turns of 45°. At 120 ft/s (~82 mph) the same calculation yields 35 seconds. For this reason flying fast or slow will not have a significant impact on the number of laps achieved. Flying faster will also be disadvantageous at the competition as it make it harder to identify the surveillance targets, harder to maintain a steady altitude, increases the chance of excessive loads on the airframe, lengthens the amount of time it will take to enter a turn, and increases landing approach speeds.

The only other option to decrease lap times is to increase the bank angle in the turns. However, the g-load on an aircraft increases significantly with bank angle and is given by the equation $1/\cos\varphi$. As this aircraft will be flying at a higher than normal weight, high g-loads will significantly increase the likelihood of structural failure. It is therefore planned to limit the bank angle in turns to a maximum of 45° to keep the loading under 1.4g. It is advantageous to try and achieve this bank angle as lower bank angles will significantly increase the lap times. For example, at bank angle of 30° and constant airspeed of 90 ft/sec it will take a minimum of 46 seconds to complete one lap compared to 33 seconds for a bank angle of 45° at the same airspeed.

5. Design Optimization/ Trade Studies/ Results

5.1. Airframe

The airframe is required to be commercially available off-the shelf (COTS) model in kit or Almost-Ready-to-Fly (ARF) form with the following specifications: wingspan – 80 to 82 inches, wing area – 1170 to 1250 square inches, constant chord – 14.75 to 15.25 inches, flat bottom (class) airfoil with 12% to 13% chord thickness ratio, and fuselage length – 62 to 67 inches. In addition, the airfoil and the majority of the airframe can not be altered. According to research of available airframes this basically limited the choices to two aircraft, the SIG Kadet Senior and the World Models Super Frontier-46. The SIG was chosen for several reasons including: a lower normal flying weight, slightly larger cabin area, larger and non-optional ailerons, and its reputation for quality construction and good flying qualities.[9]

It is planned that the final weight of the aircraft will be 14.5 lbs. or approximately 100% above the normal flying weight of 7–7.5 lbs. This increase is due primarily to the PCD batteries and heat dissipation wire. We are planning to be near the 15 weight limit for the competition in order to maximize the power output to the PCD. Engine and PCD battery location within the airframe will be movable in order to trim

[9] http://www.sigmfg.com

and balance the aircraft. This will allow the static margin (SM) to remain within safe limits and for the CG to be located at the same point as specified for normal operations of this aircraft. The aircraft can safely lift 150% of its normal weight and not run the risk of structural failure;[10] which puts the conceptual take-off weight without modifications near 11 lbs. However, this weight limit takes into account basic aerobatic training flight which is how this aircraft is normally used. These types of maneuvers can easily involve loads of over to 4 g. In contrast, it is only planned to operate this aircraft up to a maximum of 2 g, which is the load in a 60° bank. Flying the prescribed course for this competition will not require maneuvers that would impose loads on the aircraft greater than 2 g.

As a precaution the both the front and rear landing gear will be changed from the supplied configuration to stronger gear that can handle greater loads and distribute that load over a larger area. This will ensure that the increased weight of the aircraft does not cause the landing gear or airframe to fail during landing or ground operations.

Optimization of the airframe includes several modifications. One of the most important is the use of ailerons as high lift devices or flaps by using an appropriate controller capable of this type of programming. This will significantly increase take-off performance. Initially another modification was planned, moving the firewall forward. The COTS model has the firewall located in a position to accommodate an IC engine and fuel tank. An electric motor takes considerably less room than a comparable IC engine. Moving the firewall forward would allow the batteries to be located farther forward. This would allow more batteries to be carried and still retain a positive SM which is required for positive aircraft stability. However, it was found that this was not necessary as the batteries chosen for the engine and PCD can be located far enough forward to retain a positive SM. In order to provide easier access for the engine and PCD batteries, the servo tray was moved two sections rearwards in the airframe and an access panel was added to in this area to the top of the fuselage. Another change made to the airframe involves strengthening the wings, by reinforcing the wing attachment to the fuselage and by adding a strut. Other changes to the airframe include providing appropriate mounting locations and access to aircraft systems, especially the video surveillance camera. A hole in the underside of the fuselage has been made for the camera. Also, increased ventilation inside the fuselage for battery cooling will be provided by adding vents.

5.2. Propulsion and Powerplant

5.2.1. Engine

Choice of a powerplant begins with choosing an engine appropriately sized for the aircraft weight. This is done by calculating a power to weight ratio (P/W) in W/ lb. and comparing this calculation to recommended values. This calculation is shown in Eq. (1)

[10] Technical representative, SIG Mfg. Co., Inc., (641) 623-0215, telephone conversation on 11/28/2006

$$\frac{VI}{W} = P/W \tag{3}$$

Where V=voltage input to the engine controller, I=maximum current rating for the engine, and W=total aircraft weight. For Li-Po batteries the voltage used is 3.7V per cell. This is the maximum fully charged cell voltage. The nominal operating voltage for Li-Po batteries is 3.3V per cell and the minimum or cutoff voltage is normally set to 3.0V or 2.8V to avoid damaging the batteries, which can occur at lower cell voltages. The recommended range for the power to weight ratio by AXi, the engine manufacturer chosen for our design, is 70-90 W/lb for a trainer aircraft. For our design an AXi out-runner brushless motor model 4120/18 was chosen. This motor has a maximum continuous current rating of 55A and a maximum voltage for a battery made up of 6-cells in series of 22.2V. Using these numbers along with a maximum aircraft weight of 15 lbs. gives a power to weight ratio of 81.4 which is well within the recommended values. Using a 6-cell battery system also allows a lower than maximum current to be used compared with a 5-cell battery. Using a 5-cell battery in the above calculations yields a power to weight ratio of 67.8, very close to the recommended minimum value of 70. However, it is more likely that a 5-cell battery configuration would require the engine to be run closer to full throttle for longer periods of time. Out-runner brushless electric motors run much more efficiently at less than full power. The engine chosen runs at peak efficiency between 15A and 40A. With a 6-cell battery system we will be able to run the engine at peak efficiency, saving power, and providing an additional factor of safety by providing excess power should any flight be longer than 10 minutes. Figure 8 illustrates the power versus displacement curves for each of the propellers tested.

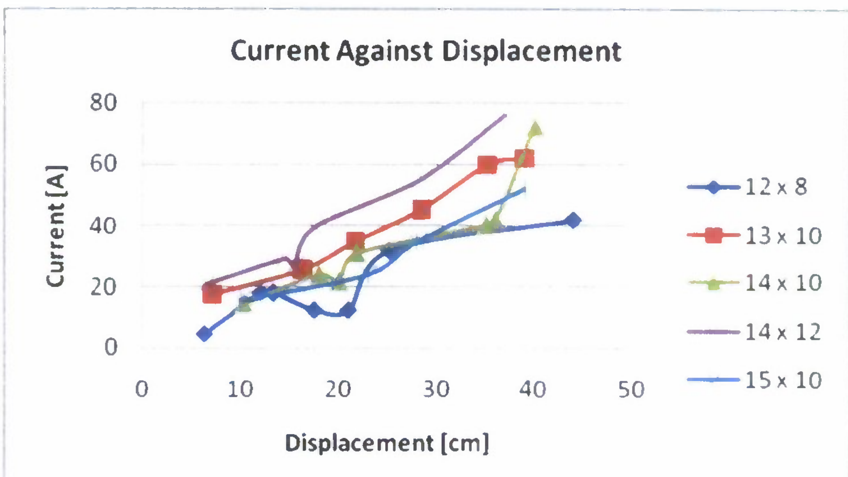


Figure 8: Figure displaying the current entering the motor and the corresponding displacement on the thrust sled.

Of the propellers tested the optimal selected propellers were the 13x10 and the 15x10. These propellers provided adequate thrust performance and handled better throughout the throttle regime than the other propellers. Other propellers have unsteady responses to a throttle input. Initial flights tests were conducted with the 13x10 propeller and performance was just as expected.

5.2.2. Engine Batteries

It was then necessary to choose a battery size. Li-Po aircraft batteries are rated in mAh or milli-amp-hours. To determine the number of milliamp hours required Eq. (2) is used.

$$A\, t\frac{1000}{60} = mAh \tag{4}$$

Where A=maximum continuous current rating of the motor in amps, t=desired flight time minutes, 1000 is the conversion factor from A to mA, and 60 is the conversion factor from hours to minutes. This formula assumes full power for a minimum flight time which provides a built in safety factor for a normal flight since full power is not usually used for the entire flight. Using A=55 Amps and t=10 min, the duration of the flight at the competition, this yields a required minimum battery rating of 9167mAh. For the engine Thunderpower batteries model TP4600-6SXL[11] were chosen. These batteries have 6 cells in series and a power rating of 4600mAh. Two of these battery packs will be used in parallel to provide a total of 9200mAh at a weight of 2.98 lbs. Solving Eq. (2) for minutes, this size battery pack will provide 2 minutes of full power at 55 amps and an additional 11 minutes at 40 amps, providing an ample flight time and factor of safety, for example when a go-around and second landing attempt is needed.

Next it is necessary to check the maximum continuous discharge, or C rating, of the batteries. C rating is determined by Eq. (3)

$$\frac{A}{cap} = C\,rating \tag{5}$$

Where A=maximum continuous current and cap=Ah capacity of the battery. For our design A=55/2 per battery pack and cap=4.6. This yields a continuous discharge rating of 6C which is well below the maximum value for the chosen engine batteries packs which are rated at 22C.

One factor related to the competition for the batteries is the *charge* C rating. Li-Po batteries are charged at 1C. This means that it will take 1 hour to charge a fully discharged Li-Po battery pack. At the competition, if we are required to fly another sortie before all four engine and PCD battery packs from a previous flight can be charged, we will need to have an additional set of batteries.

5.2.3. PCD Batteries

Two primary factors controlled the design and selection of the PCD batteries. First, the PCD system is required to operate at 28V. As a battery is discharged the voltage drops. In order to maintain 28V without a heavy step-up transformer and related electronics the PCD battery pack must always be above

[11] http://thunderpower-batteries.com/Li-PolyBatteries.htm

28V. This necessitated the selection of a 10 in series Li-Po battery pack. This yields a cut off voltage for this battery system of 28 to 30 V allowing for a full discharge for maximum power output. Second, the weight of this battery pack and the required PCD designed to handle the power output must be taken into account. All the other on-board aircraft systems are required. Only the PCD system can be varied to remain below the 15 lb limit once the propulsion system has been selected.

The PCD batteries were chosen by using Eq. (2) solved for A. This associated weight of this size battery system was then found, and finally the total weight aircraft could be calculated. Using this procedure iteratively two Thunderpower battery packs, model number TP4000-10S2PL[12] were chosen for the PCD. These batteries packs are rated at 4000 mAh giving a total power output of 8000 mAh or 47.6 amps for 10 minutes at a weight of 3.60 lbs. At 28V this yields a total power output of 1333 W. This figure is below our initial estimate of 1680 W at a lighter weight due to two factors. First, a custom battery configuration was not available which would have provided more power and second, a mistake was made when calculating battery configurations and associated available mAh leading to an under-estimated total battery weight.

The C rating for the PCD batteries was also computed. For our design the maximum continuous current is A=47.6/2 per battery pack, and cap=4.0. This also gives a continuous discharge rating of 6C. This is well below the maximum value for the PCD battery packs which are rated at 12C.

5.2.4. Electronics

The electronics in the aircraft and the power supplied to those systems are described in this section. First the configuration of the aircraft control transmitter (TX) needed to be determined. The rules for the competition require a TX that includes synthesizer frequency module 72MHz band, with PCM modulation. This is a fairly sophisticated TX and in addition we wanted to have the capability of using the ailerons as flaps commonly called flaperons. This required at least a 6-channel TX in order to control both ailerons separately, rudder, elevator, motor, and PCD. A 9-channel PCM transmitter with flaperon programming was chosen from Futaba, model number 9CAPS with receiver (RX) model number R319 DPS.[13]

Next servos were chosen. The rules required minimum 50oz-in torque servos and two choices are available, analog and digital. Additionally, they can be run at either 4.8 or 6 V. Digital servos require more power but they are more accurate and have better feedback control. This makes it easier to fly in gusty wind conditions where the controls can be deflected from the desired position. As a result digital servos

[12] http://thunderpower-batteries.com/Li-PolyBatteries.htm

[13] http://www.futaba-rc.com/radios/futk75.html

from Futaba, model S3151[14], were chosen. Figure 9 is a picture of the constructed voltage regulator before it was covered.

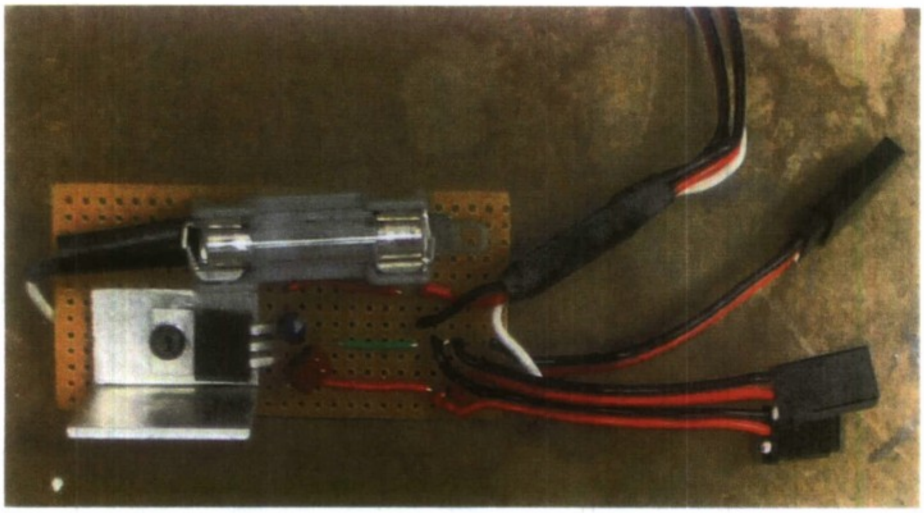


Figure 5: Picture of the constructed voltage regulator board.

Next the battery for the electronics needed to be decided. Running the servos at 6.0 V also provides more power to the control surfaces compared to running them at 4.8 V. A 6.0 V servo system is an advantage in our situation since the weight of the aircraft will be above normal. This could cause sluggish flight characteristics that would be counteracted by the more powerful controls. In addition, both the video system and the telemetry system require at least 5.0 V for operation. Since as battery is discharged the voltage drops, in order to maintain the required 5.0 V it was decided to use a 6.0 V battery for all the electronics.

Since a 6.0 V battery system was chosen it is required to build a voltage regulator for the video system. The receiver and the telemetry recorder and TX have their own voltage regulators. This voltage regulator is a simple circuit based on a LM2940 chip along with a single capacitor. One circuit is adequate to supply the required current load while minimizing the power lost to heat.

Battery size and type then needed to be determined. The radio RX and servos according to the rules and must have a 1000mAh minimum NiCad or NiMH battery. NiMH batteries are lighter but they when under load are more likely to have a significant voltage drop. Eq. (2) was then used to determine battery size for both the RX and servos and the remaining electronic systems. Electrical load requirements for these two systems are outlined below in Table 4.

[14] http://www.futaba-rc.com/servos/digitalservos.html

Table 4: Electronic power requirements per component

Aircraft Control		Electronics	
Part Description	**Power (mA)**	**Part Description**	**Power (mA)**
Receiver	16	Camera	200
Control surface servos (4)	1600	Video Transmitter	40
Throttle control servo	400	Telemetry Device	70
Total	2016	Telemetry Transmitter	40
		Voltage Regulator	28
		PCD control electronics	400
		Total	778

Using these power requirements and Eq. (2) it was found that the RX and servos required only a 336 mAh battery and the remaining electronics only 188 mAh. NiCd and NiMH batteries are commonly available in 1200 mAh size, however, NiMH batteries are lighter and a 1650 mAh battery is the same size and weight. Therefore a 1650 mAh NiMH battery was chosen to power all the electronics in the UA. This greatly oversized battery allows for a dedicated 1000 mAh to be allocated to the RX and servos with plenty of power remaining to account for the digital servos running at 6 volts and more than enough power to maintain a minimum constant 5 volts for the remaining electronics.

The camera and telemetry systems were chosen largely on recommendations from the Air Force and the reputation of the reliability of these systems in airborne applications. The telemetry system requirements for this competition are also fairly complex, which limited the choices available. For the competition the telemetry system will need to track altitude, airspeed, PCD voltage and current. In addition the telemetry system will be used to track engine battery voltage and current, engine RPM and cabin temperature in order to provide optimal control of the aircraft and maintain safety. For the telemetry system the Seagull Pro Flight System from Eagle Tree Systems was chosen. For the video surveillance system the Brown Bag Kit from Black Widow AV[15] was chosen.

[15] http://www.blackwidowav.com/brownbagkits.html

5.2.5. PCD

The PCD is made up of two primary components, a voltage or current regulator and a heat dissipation device. Once a battery size and associated maximum power output was determined design work was done to provide these components that could handle the proposed power output.

Initially the Air Force was to provide the PCD. Their custom design used a MOSFET transistor for voltage control. A MOSFET efficiently regulates voltage by rapidly turning on and off the circuit. This also keeps the heat generated by the voltage regulator to a minimum compared with other voltage regulator designs, However, while the Air Force design was controllable from the ground it had no automatic feedback capabilities and could only handle a maximum of 500 W. As a result of our, and possible other similar proposals, the Air Force changed the competition so that each team is required to provide their own PCD. The Air Force in an email to us stated that they were "very interested" to see how we were going to handle the high power we proposed and the associated heat dissipation problem. The simplicity and high reliability of the design created for our PCD will easily be able to handle the power available from the PCD batteries and, we believe, will provide our team with a huge advantage at the competition.

Instead of using a custom circuit design that would have difficulty handling large amounts of power it was realized that a high efficiency airborne system for high power voltage control already existed. Tested and proven brushed motor controllers are commonly available that can handle up to 5400 W. These motor controllers are set up for easy and full control from the ground using the normal R/C transmitter and receiver set-up. In addition, it was realized that these controllers use servo control circuitry that provides automatic feedback for the voltage control circuit. This will allow us to set the output of the PCD from the ground without having to adjust it for varying conditions during the flight. For PCD control we will be using a Jeti JES 80 motor controller which is capable of complete voltage control from 0 to 36 V at up to 80 amps. In addition, during ground testing it was found that this setup will provide excellent voltage regulation at 28 ±0.1 V.

For the heat dissipation we are using heating resistance wire such as those used in small appliances and commercial heaters. No fin or other heating element support will be used as in the Air Force design.[16] Instead the wires will simply be held in place between the fuselage and outer portion of the horizontal stabilizer of the aircraft. This will provide a large amount of air flow around wire for maximum cooling, keep the heat generated well away from the aircraft, and will have a minimal effect on aircraft aerodynamics compared to a fin design. Only the size and length of wire needed to be determined along with a mounting system to protect the airplane from the heat generated.

For this application it will be necessary to have the heating element wire remain relatively flexible due to the forces applied to the aircraft and wire during normal flight conditions. As a result Nikrothal 80, a

[16] Email received from AFRL containing wiring diagram, specifications, and photographs of AFRL PCD design, January 19, 2007

nickel-chromium heating wire available from Kanthal that remains flexible at elevated temperatures *and* after cooling was chosen. Since the PCD circuit is to run at a constant 28 V and the current we calculated can be delivered from the selected batteries is 47.6 continuous amps for 10 minutes, it is required to set the heating wire resistance at 0.588 Ω according to Eq. (4) where V=voltage, I=current, and R=resistance.

$$\frac{V}{I} = R \tag{6}$$

To calculate the wire size and length, recommendations from the Kanthal Heating Alloys handbook were used. For an unsupported heating element in forced air convection Kanthal recommends a surface load of 45-50 W/in^2 which yields a wire temperature of approximately 600° F. For our application the equations given by Kanthal for determining surface load were reduced to Eq. (5) where I=current, C_t=wire temperature resistance factor, p=surface load, and SAR=surface area to resistance ratio.

$$\frac{I^2 C_t}{p} = SAR \tag{7}$$

Using I=47.6 A, C_t=1.03 from Kanthal tables for Nikrothal 80 at 572° F, p=45, the minimum SAR for our application is 46.7 in^2/ohm. Nikrothal 80 at AWG size 10 has a maximum SAR rating of 60.8 in^2/ohm at 68° F, well above what is required for our application which should yield cooler operating temperatures. (The next smaller gage wire, AWG 11, has a SAR rating of 42.9 in^2/ohm.) Nikrothal 80, AWG 10 has a resistance of R_w=0.0631 ohms/ft. Eq. (6) gives the heating element wire length required.

$$\frac{R}{R_w C_t} = L \tag{8}$$

Using R=0.588 Ω, R_w=0.0631 ohms/ft, and C_t=1.03 yields a heating element wire length of L=9.05 ft or 108.6 in. This length of PCD heating element wire will be configured in 4 lengths of wire 27.15 inches long run between the side of the fuselage and the outside front edge of the horizontal stabilizer. This wire will be held in place by springs which will help control sag of the wire as it is heated, provide damping from dynamic forces, and act as a heat sink to dissipate heat before it reaches the airframe. These springs will be held in place by a threaded rod attached to the airframe at the appropriate locations.

6. Detail Design

This sections interprets the preliminary design section and presents it as the final design. A table summarizing all the inclusive parameters will be discussed here. More detailed sections below are only for parameters that need additional explanation, otherwise they are simple displayed in the summarized table.

6.1. Performance

6.1.1. Aerodynamic Parameters

Lift coefficient was determine through a combination of methods. As stated above three different methods were used to predict the aerodynamic performance. The wind tunnel tests were used to derive the drag polar for the airframe, this was then scaled up to the full-model dimension to be used for a variety of performance measures. Equation 9 is the estimated drag polar:

$$C_D = 0.245 + 0.7356 C_L^2. \qquad (9)$$

Numerical analysis was done on the airfoil to bring about the maximum lift coefficient. Similar Reynolds and Mach Numbers were used to arrive at a value of max C_L =1.26. Figure 10 is a plot generated to validate the result.

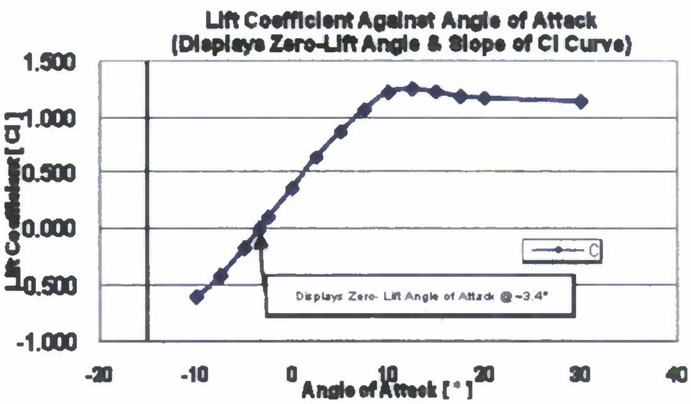


Figure 10: Plot of lift coefficient against angle of attack for similar airflow to that predicted at the competition.

Lift-over-Drag (L/D) ratios were calculated from both the drag polar and the experimental tunnel data. The resulting maximum L/D was around 9.03. Maximum rate of climb is determine from the excess power available from the propulsion system when the L/D ratio is at its maximum and is describe by Equations 10-12.

$$\dot{h} = \frac{P_A - P_R}{W} \qquad P_A = \eta P_{shaft} \qquad P_R = D = 0.5\rho V^3 S C_D \qquad (10\text{-}12)$$

Where S is the planform area of the wing. The stall speed is based on $C_{L max}$ and the geometry of the airplane, Equation 13 related the stall speed to the aerodynamic performance.

$$V_{stall} = \sqrt{\frac{2W}{\rho S C_{L,max}}}, \qquad (13)$$

The maximum speed the airplane can obtain is based off the maximum rotational speed of the propeller and thus the power available. The power required increases as the airplane increase velocity. There is a limit to the amount of excess power the engine can generate, thus the airplane's acceleration decrease to eventually zero. The resulting maximum speed is around 70 mph, but the pilots will purposely limit the airplane to a speed of around 40 mph to maintain stability perform safe maneuvers. The takeoff distance uses an approximation to the takeoff forces. The resulting takeoff distances for both on short grass and pavement are 95.89 and 71.97 ft, respectfully.

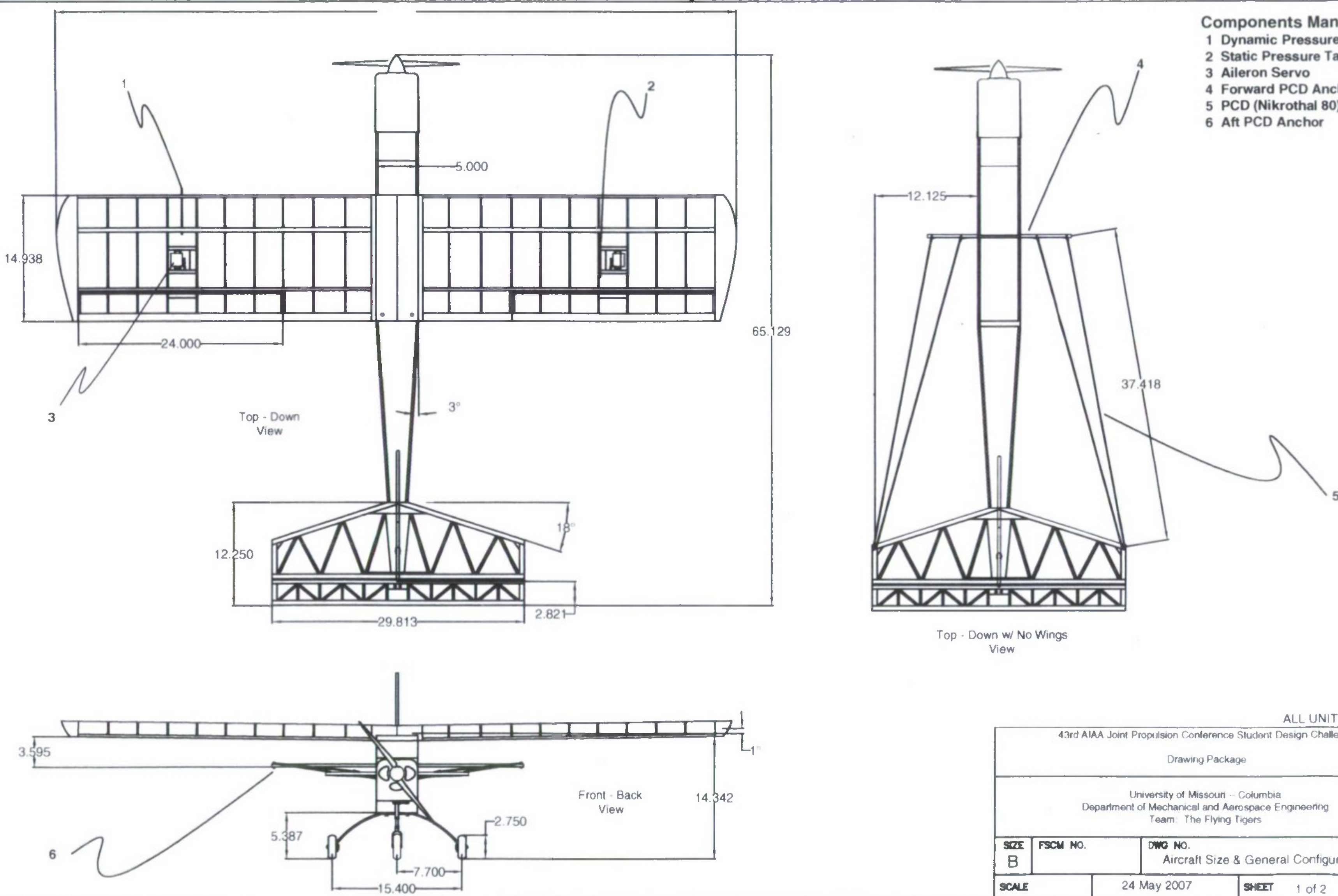
6.2. Weight and Balance

The goal of this task is to keep the originally designed CG in the same position as when it is fully loaded, it is safer to error on the side of nose-heavy. The airplane becomes unsteady if the weighted CG is aft of the original CG. In order to quantify the moment balance about the original CG, weights and approximate moment arms need to be developed. Therefore Table 5 is a detailed list of the weights of the airframe's components. The detailed drawings offer a good means of approximating the moment arm about the approximate CG of the particular component. It should be noted that these values are approximate and are not measured to the exact component CG. Wiring also throws off the weighted CG because the wire CG is difficult to determine.

Table 5: Weight and balance estimate. The result displays that the plane is slightly nose-heavy, which is more desirable than tail-heavy.

Component	Includes	Weight [lbs]	Balance [in]	Resulting Moment [lb-in]
Airframe	fuselage, landing gears, wheels, stabilizers, windscreen, cowl	3.1023363	8.4396	26.18247744
Wing Assembly	L end R wings, dihedral bar, telemetry pitot tubes	2.0125	2.3115	4.65189375
Propulsion Hardwere	nose cone, 15 x 10 propeller, retaining hardwere	0.153241	-17.0468	-2.612268679
Motor Assembly	motor, bracket, hardware	0.7	-14.9329	-10.45303
Motor Batteries	-	2.9	-7.516	-21.7964
Motor Controller	-	0.11368	-13.0769	-1.486581992
70 Amp Fuse	-	0.016556291	-1.0989	-0.018193709
Servos	extension wire, connections	0.375	11.9974	4.499025
Pushrods	-	0.0101637	27.864	0.283201337
RX	-	0.0838852	-8.9366	-0.749648478
Wiring	-	0.179690949	6.4265	1.154783885
RX Battery	-	0.31346578	-5.3844	-1.687825146
Camera Assembly	camera and antenna	0.142384106	6.1101	0.869981126
Telemetry Herdwara	-	0.052980132	3.5301	0.187025164
Telemetry Antenne	-	0.03863134	3.8326	0.148058474
Voltage Regulator	-	0.062251655	7.3862	0.459803174
Sensor Assembly	voltage and current sensors	0.002207506	0.5935	0.001310155
PCD Wire	weight per unit length	0.37012987	25.059	9.275084416
60 Amp Fuse	-	0.016556291	13.3029	0.220246689
PCD Controller	-	0.10816777	11.9814	1.296001319
PCD Batteries	-	3.5	-3.2893	-11.51255
	Total Gross Weight [lb]	**14.25382789**	-	-
			Total Moment [lb-in]	**-1.087606078**

A convention is adopted to standardize the moments produced. If the component CG is forward of the desired original CG then the moment produced is negative, in that the resultant force causes the nose to pitch down. This is likewise opposite for components center aft of the original CG. Table 7 is the final summarized performance table, located after the drawings, and is the conclusion to the detailed design section of the report.



Components Manifest 1/2

1 Dynamic Pressure Tap
2 Static Pressure Tap
3 Aileron Servo
4 Forward PCD Anchor
5 PCD (Nikrothal 80)
6 Aft PCD Anchor

ALL UNITS IN INCHES

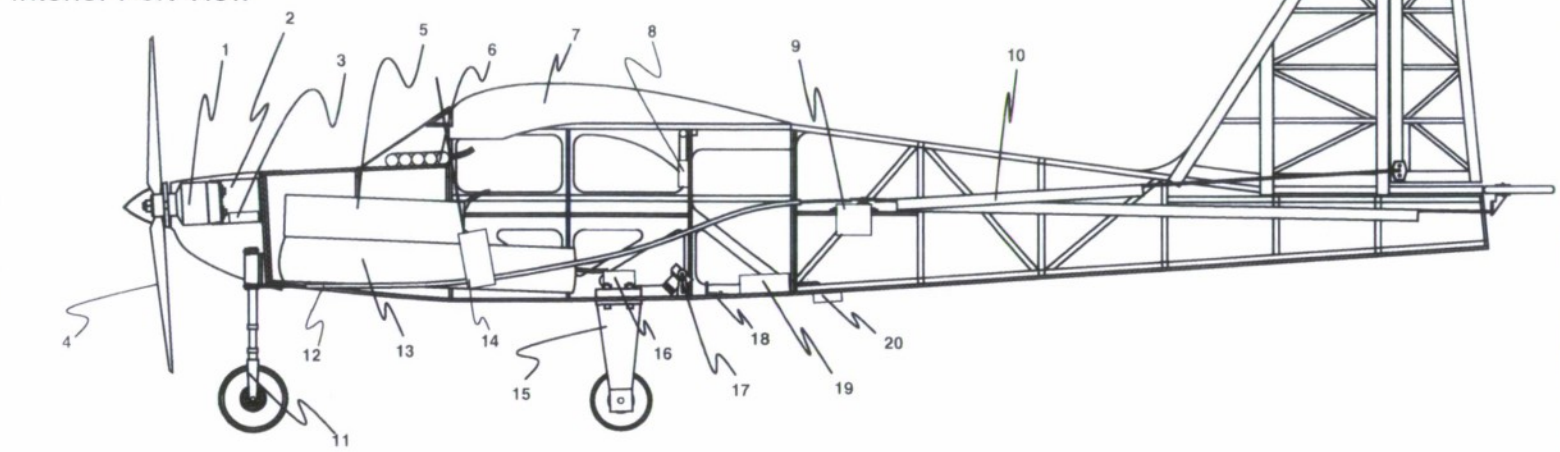
43rd AIAA Joint Propulsion Conference Student Design Challenge

Drawing Package

University of Missouri -- Columbia
Department of Mechanical and Aerospace Engineering
Team: The Flying Tigers

SIZE	FSCM NO.	DWG NO.	REV
B		Aircraft Size & General Configuration	
SCALE	24 May 2007	SHEET 1 of 2	

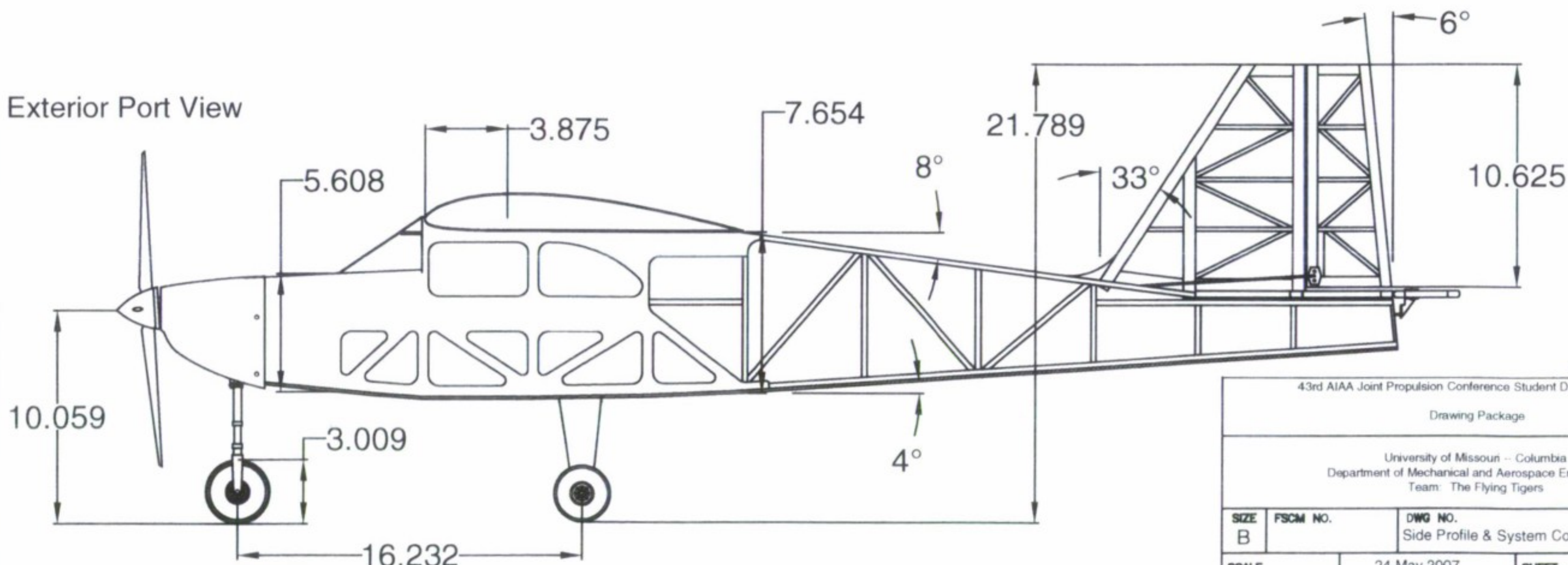
Interior Port View



Components Manifest 2/2

1 Motor
2 Motor Mount
3 Motor Speed Controller
4 Propeller
5 Lithium Polymer Motor Batteries (2 packs)
6 NiMH Receiver Battery
7 Wing Root
8 Video Transmitter
9 Elevator and Rudder Servos
10 Elevator and Rudder Pushrods
11 Nose Gear
12 Nose Gear Flexible Pushrod
13 Lithium Polymer PCD Batteries (2 packs)
14 Telemetry Antenna
15 Main Gear
16 Telemetry Recorder
17 Camera
18 5V Voltage Regulator
19 Receiver
20 PCD Voltage Controller

Exterior Port View



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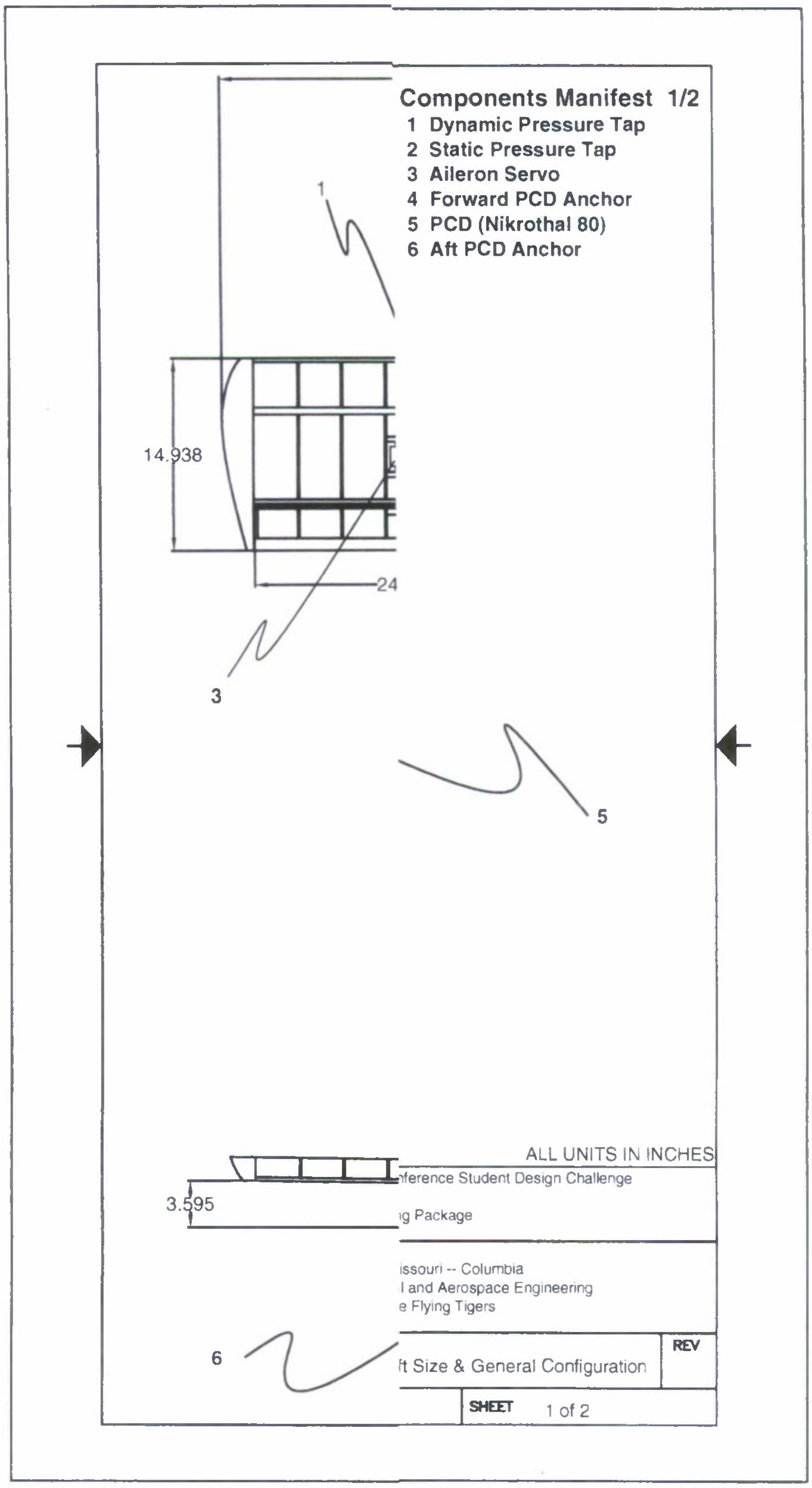
43rd AIAA Joint Propulsion Conference Student Design Challenge Drawing Package			
University of Missouri -- Columbia Department of Mechanical and Aerospace Engineering Team: The Flying Tigers			
SIZE B	FSCM NO.	DWG NO. Side Profile & System Component Layout	REV
SCALE	24 May 2007	SHEET 2 of 2	


Components Manifest 1/2
1 Dynamic Pressure Tap
2 Static Pressure Tap
3 Aileron Servo
4 Forward PCD Anchor
5 PCD (Nikrothal 80)
6 Aft PCD Anchor
1
14.938
24
3
5
3.595
6
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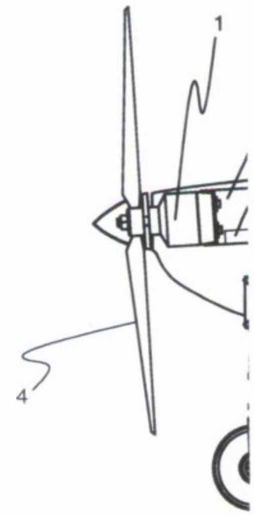
REV
SHEET 1 of 2

Interior Port View



Components Manifest 2/2

1 Motor
2 Motor Mount
3 Motor Speed Controller
4 Propeller
5 Lithium Polymer Motor Batteries (2 packs)
6 NiMH Receiver Battery
7 Wing Root
8 Video Transmitter
9 Elevator and Rudder Servos
10 Elevator and Rudder Pushrods
11 Nose Gear
12 Nose Gear Flexible Pushrod
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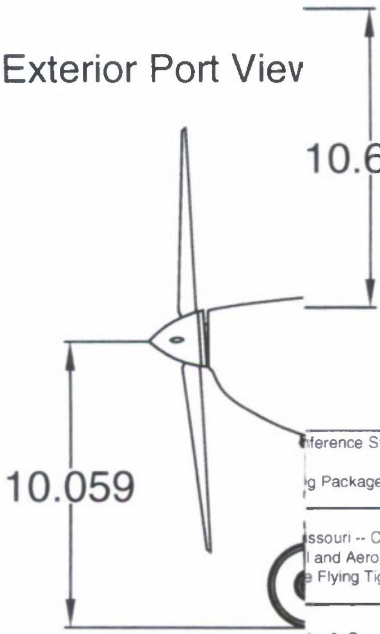
15 Main Gear
16 Telemetry Recorder
17 Camera
18 5V Voltage Regulator
19 Receiver
20 PCD Voltage Controller

Exterior Port Viev



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le & System Component Layout	REV
	SHEET 2 of 2

Table 6: **Detailed design summary table.**

Geometry		Performance		
Length [in]	~65.129	Max Lift Coefficient	1.26	
Span [in]	~80	Max L/D	9.03	
Wing Area [in^2]	1180	Drag Polar	0.245 + 0.7336	
		Maximum Rate of Climb [ft/s]	-	
		Stall Speed [ft/s]	35.0139	
		Maximum Speed [ft/s]	64.4306	
		Takeoff Distance [ft]	Short Grass	95.8901
			Pavement	71.9719
Weight & Balance		**Systems**		
	Reference Table 5	Radio	Futaba 9C Super PCM/PPM	
		Receiver	Futaba R319 DPS	
		Servo	Futaba S3151 x 4	
		Battery	Motor	Thunder Power TP4600-6SXL (2 in parallel)
			PCD	Thunder Power TP4000-10S2PL (2 in parallel)
			Receiver	6V 1650 mah NiMH
		Motor	Model Motors AXi 4120/18 BL electric motor	
		Controller	Jeti ADVANCE 70 Opto Plus	
		Propeller	APC 13 x 10	
			APC 15 x 10	
		Video	Black Widow200mw Brown Bag Kit	
		Telemetry	Eagle Tree Systems Pro Wireless w/ expanders	
		PCD	Controller	Jeti Brushed JES 80 Amps
			Wire	Kanthal tables for Nikrothal 80

7.0. Testing Summary

Testing is the only true way to understand if any design is destine for success or failure. Along the design timeline there are tests that were done to verify certain uncertainties in the design. For example, deciding which propeller to use is based off static and dynamic testing. Testing schedules can be referenced to Figure 2. Due to space constraints on the report, full detailed checklist could not be include, but if there was space, they would consist of events that need to occur to give a green light for tests. In the case for Li-Po batteries, the battery packs need to be broken in slowly before they can be discharge to their full capacity, this is mainly a factor of safety. We are now at the stage of being able to discharge the batteries to their full capability. If this were to be done again, we would start the testing sooner to give us more time to break the batteries in. Additionally, a better communication feedback from the group members would certainly help the team's overall performance. If we were to pass anything down to future design teams if would be that everything that you think you can do, plan on twice as long for it actually to get done. There are so many small little things that you simple cannot account for. Another thing would be to never assume that your planning design will work, test everything to make sure, assemble the plane and put everything in it to make sure. The bottom line is the more testing that is done the more confident you can be in your design, they are directly proportional.